Semi-Autonomous Spacecraft On-Board Ephemeris Propagation

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The increasing high demands on space systems together with the current national and international economic realities necessitate a new vision for space missions. The National Aeronautics and Space Administration (NASA) new vision concentrates on small efficient spacecraft and new mission operations concepts which include low cost launch vehicles and low cost mission operations. The new mission operations concepts should provide automation and selected migration of operation functions to the spacecraft. Autonomous navigation, spacecraft self-health analysis and correction, and on-board sequence generation and validation are examples of these automation concepts. This autonomy enables significant reduction in operations intensity and staffing and network utilization.

Autonomous navigation can be accomplished by automating and migrating orbit · determination, trajectory propagation and maneuver design to the spacecraft. This paper is concerned only with autonomous on-board trajectory propagation. This function is extremely important not only because it provides support to other spacecraft subsystems like attitude determination and control, and antenna/instrument pointing but also because it is a key element in the other autonomous navigation functions, namely orbit determination and maneuver design.

Traditionally, there are two approaches for ephemeris propagation. One approach is through a step-by-step numerical integration (special perturbation) which implements accurate force models and provides a precise trajectory with the disadvantage of slow computation. The other approach is through analytical expansion and integration of the equations of variations of orbital parameters (general perturbation) which implements less accurate force models and provides approximate solutions and has the advantage of fast computation. A method which combines the advantages of these two approaches is called the semianalytical method in which numerical integration is performed only on the equations for the mean rates of the orbit parameters with large step size, and then the solution is added to the short-period variations (Ref 1 and 2).

In this paper two strategies for autonomous on-board ephemeris propagation are presented. One strategy rectifies, when needed, a simple cm-board (on-line) ephemeris using an accurate ground (off-line) ephemeris. This method is called rectification. The other strategy applies the lessons learned from TOPEX/POSEIDON on-board ephemeris representation to build a data compression device like the Fourier Power Series (FPS) for a wide range of orbits. A trade-off study is conducted in this paper to relate accuracy requirements with the semi-major axis and eccentricity in such a way to minimize the frequency of ephemeris uplinking. In the following, the two strategies are explained in more detail with some preliminary results.

1. The Method of Rectification

With the rapid development in microprocessor technology and on-board processing, new and autonomous tools of ephemeris generation will become commonplace. For instance, the NASA Standard Spacecraft Computer (N SSC- 1) used on TOPEX/POSEIDON has the capability of 200K operations/see. In more recent developments, as in the Cassini 'and Pluto fly-by missions, the on-board processors have the capability of multi -mega operations/see. With these processors on-board propagation with simple force models are feasible.

In the method of rectification two ephemeris propagators are used. The on-line propagator which runs in real-time on-board the spacecraft uses a simple force model and implements the method of Ref(1) or (2). Reference (1) presents a semianalytical propagator in Poincare elements using the generalized Lie-Hori method where the equations for the mean rates are numerically integrated with a large step size and then added to the short-period variations. The perturbation method of Ref (2) is also used in this paper for comparison. It also integrates the equations for the mean rates and adds the short-period terms in some set of equinoctial elements. Both on-line methods use a simple force model (geopotential up to J2**2 only). The advantage of the methods of Ref(1) and (2) is their speed. The step size in both methods increased by a factor of a thousand for TOPEX/POSEIDON compared with the numerical integration of the exact equations of the simple model. The off-line propagator uses a complex force mode] and integrates the exact equations step-by-step. It is used to update the on-line solution when needed using the method of rectification (Fig 1). Both the on-line and off-line trajectories use the same initial conditions. As the size of the along-track difference between the two trajectories increases to an unacceptable difference, a new time and set of initial conditions based on the off-line ephemeris, consisting, of only seven parameters, should be uplinked. Figures (2) and (3) show a typical result of this approach. The on-line solution used here is based on Ref (1) and the off-line model includes 9x9 geopotential, drag, and solar radiation pressure. The figures show the maximum nadir pointing error if rectification is done after 20 and 30 days for a wide range of' semi-major axis and eccentricity. They show that the pointing error computed by the on-line trajectory generally decreases with height to an absolute minimum near the 12-hour orbit, then increases as the altitude increases to the geostationary height. At very low altitude the limited J2 geopotential and lack of drag effect in the on-line trajectory are responsible for this high pointing error. The authors intend to add the J3 term to Ref(1) in an attempt to increase the accuracy of the on-line ephemeris. Although the pointing error above the 12-hour orbit is reasonably good, the increase in the error as height increases is due to the effect of luni-solar perturbations.

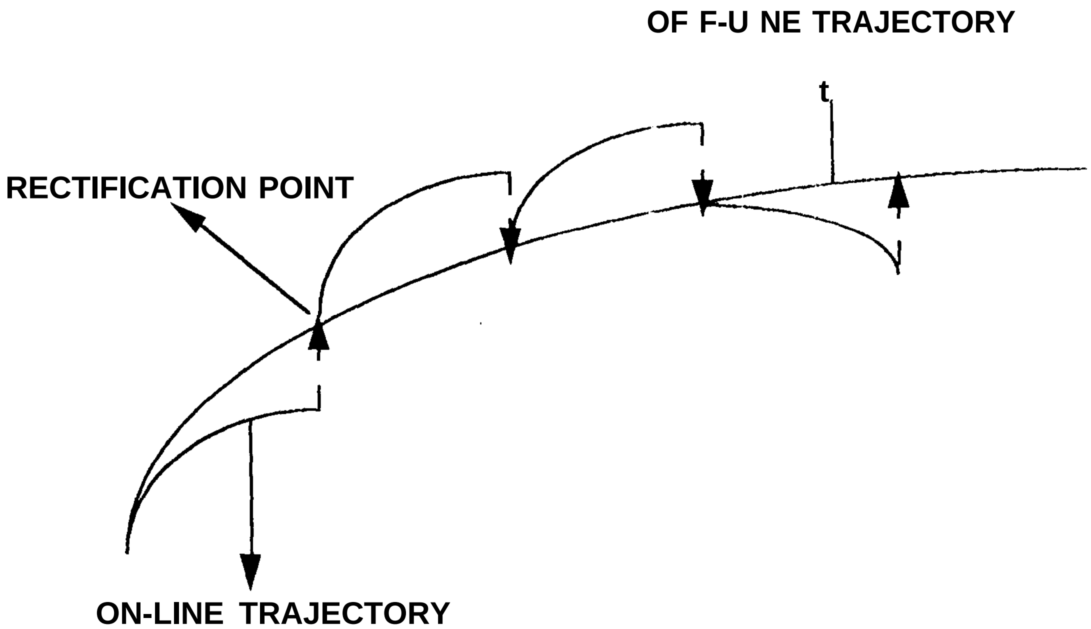


Fig (1)
Rectification of On-line Trajectory

2. Ephemeris Compression

TOPEX/POSEIDON uses the FPS as an ephemeris compression device to uplink the ephemeris to the spacecraft. A 42-coefficient FPS representation is used for each of the six Cartesian state vector components of the accurate ground ephemeris. This ephemeris load is uplinked to the spacecraft weekly. This strategy applies the lessons learned from TOPEX/POSEI DON on-board ephemeris representation to build a data compression device like the FPS at a wide range of orbits. A trade-off study is conducted in this paper to relate accuracy requirements with the semi-major axis and eccentricity in such a way to minimize the frequency of ephemeris uplinking. By minimizing the frequency of uplinking we increase the degree of autonomy. Figure (4) shows a typical result of this approach for 20-day and 30-day fits. The pointing error drops rapidly as the height increases from about several hundred kilometers to about the Topex height. Then the effect of drag decreases so that the accuracy of the fit is almost constant and the length of the fit could be even extended more than 30 clays.

References:

1. Salama, Ahmed. "A Semi-Analytical Method Based on t h e Generalized Lie-Hori Approach" AAS/AIAA Astrodynamics Conference, Vail, Colorado, August 1985.

2. Konopliv Alex," A Perturbation Method and Some Application ," Celestial Mechanics, 47,pp.305-320, 1990 .

20-DAY RECTIFICATION

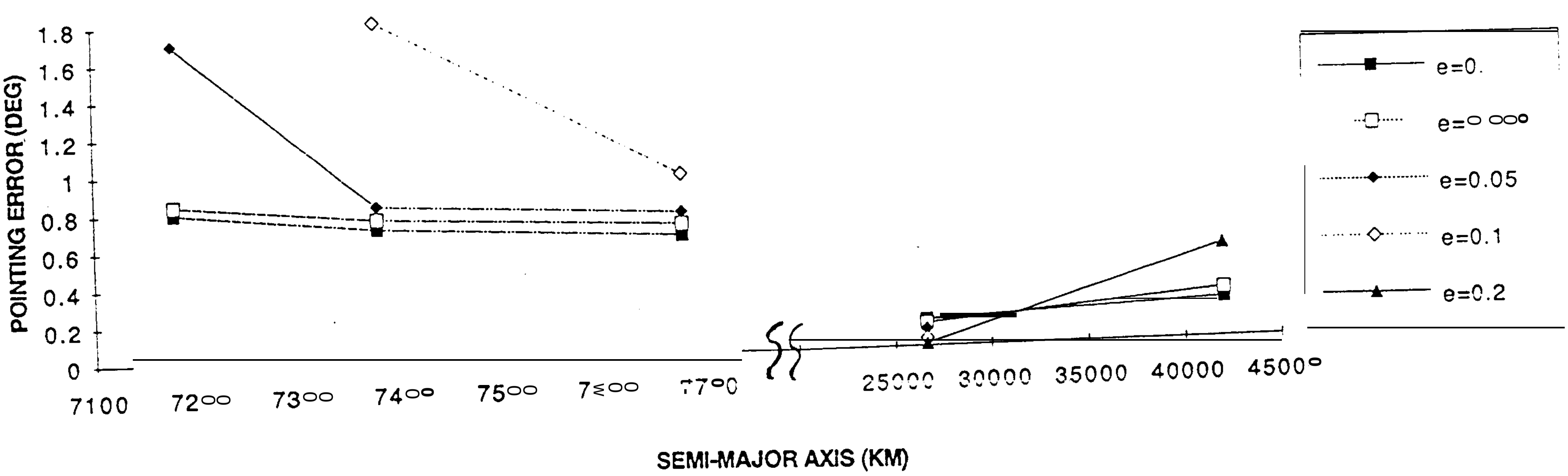
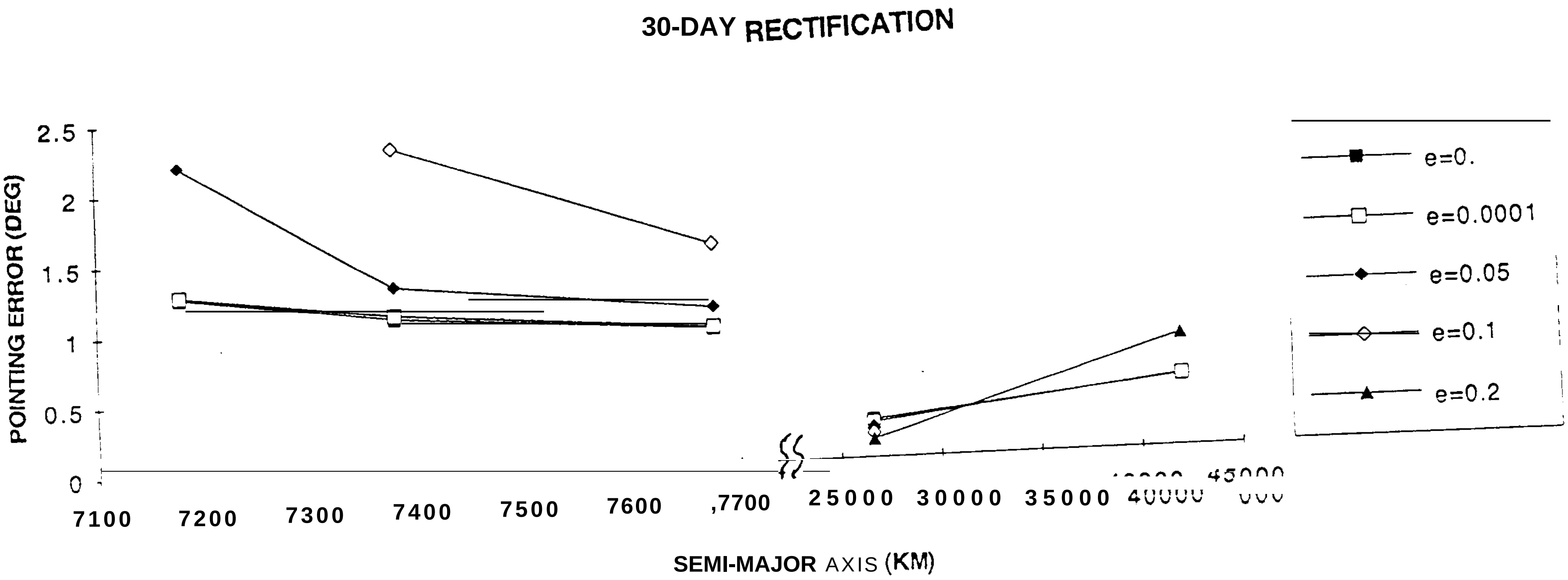
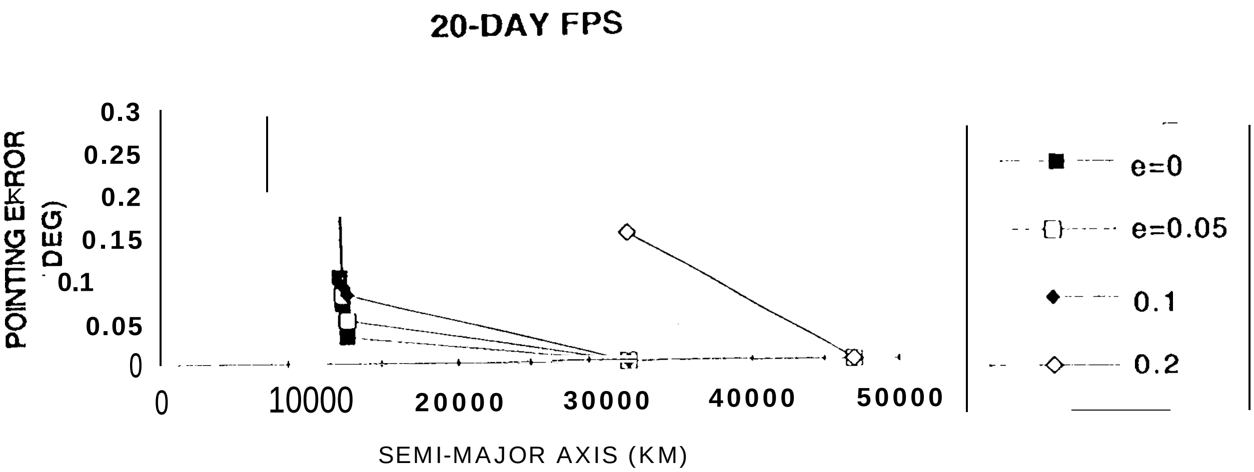


Figure (2)



30-DAY RECTIFICATION
POINTING ERROR (DEG)
2.5
2
1.5
1
0.5
0
7100
7200
7300
7400
7500
7600
7700
25000
30000
35000
40000
45000
SEMI-MAJOR AXIS (KM)
e=0.
e=0.0001
e=0.05
e=0.1
e=0.2

Figure (3)


20-DAY FPS
POINTING ERROR (DEG)
0.3
0.25
0.2
0.15
0.1
0.05
0
0
10000
20000
30000
40000
50000
SEMI-MAJOR AXIS (KM)
e=0
e=0.05
0.1
0.2

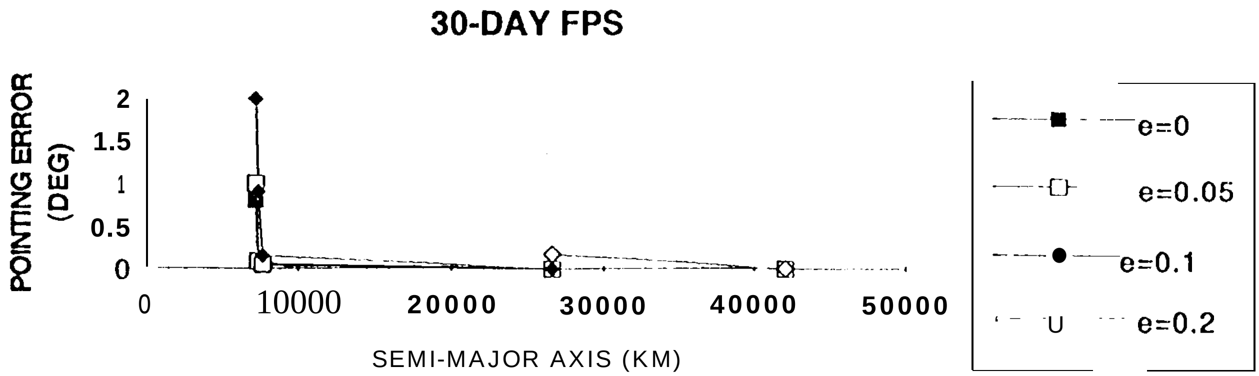

30-DAY FPS
POINTING ERROR (DEG)
2
1.5
1
0.5
0
0
10000
20000
30000
40000
50000
SEMI-MAJOR AXIS (KM)
e=0
e=0.05
e=0.1
e=0.2

Figure (4)